Bellcomm

955 L'Enfant Plaza North, S.W.
Washington, D. C. 20024

B71 06039

date: June 23, 1971

to: Distribution

from: E. M. Grenning

subject: A Possible Lunar Logistics Mode
Case 105-4

ABSTRACT

A mode of operation is described for repetitive logistics operations between an earth orbit space station or propellant depot and a polar lunar orbit space station.

The mode selected is characterized by a two month periodic recurrence of a set of earth departure and lunar departure windows. The duration of the five earth departure windows varies from 3 hours to 24 hours while the four lunar departure windows are from 6 hours to 35.5 hours in duration.

The vehicle that operates in the derived logistics mode is a nuclear shuttle with an average specific impulse of 795 sec. and a mass fraction of .75. Based on the calculated translunar and transearth velocity changes and the corresponding payloads of 119 K lbs. and 0, respectively, the shuttle gross weight at the beginning of the mission is 352 K lbs.

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MEMORANDUM FOR FILE

1.0 INTRODUCTION

The purpose of this memorandum is to illustrate the characteristics of a lunar logistics mission mode including the frequency of mission opportunities, launch window durations, and cislunar shuttle size. Pursuant to these objectives it is convenient to divide the analysis into four sections: mission profile; mission geometrical considerations; mission velocity requirements; and vehicle sizing.

2.0 MISSION PROFILE

A lunar logistics mission is assumed to begin with the departure of the cislunar shuttle during an earth departure window from an earth orbital space station or propellant depot to which it returns at the end of the mission. After translunar flight the shuttle performs a three impulse maneuver to rendezvous with a lunar polar orbital space station employing an intermediate elliptic orbit for plane change as needed. After a period of time which is at least sufficient to allow transfer of personnel and cargo between the station and shuttle, the trip back to earth begins at the first available lunar departure window. The transearth flight is initiated with a three impulse maneuver using an intermediate, elliptic orbit for plane change as needed. At earth arrival the shuttle maneuvers into a circular earth orbit which is coplanar with, but of higher altitude than, the earth orbital facility to obtain the correct phasing for terminal rendezvous. A plane change maneuver is not needed since the shuttle arrives at earth at the time of occurrence of an earth arrival opportunity. When the correct phasing is achieved the shuttle performs a Hohman transfer to the facility in the lower altitude orbit and then terminal rendezvous.

3.0 MISSION GEOMETRICAL CONSIDERATIONS

Figure 1 illustrates the arrangement of the earth departure and lunar orbital planes with respect to the earth's equator and poles. A zero plane change earth departure or arrival opportunity occurs whenever the target line between the earth and moon lies in the earth orbit plane (Reference 1). This target line must lead the moon or earth by the flight time of the translunar or transearth trajectory respectively. As the moon orbits about the earth with an angular velocity of $\omega_M = 13.2°/\text{day}$ the target line rotates in the lunar orbital plane at the same rate, and lies in the earth's orbit plane when it passes through the intersection of the two planes. But the intersection of the two planes rotates with respect to an inertial frame of reference due to the nodal regression of the earth orbit. The relationship between nodal regression rate (ω_N) and orbital inclination and altitude is given in Figure 2, from Reference 2.

Figure 3 illustrates the motions of the earth-moon line and the line of intersection of the planes in the lunar orbital plane. For a periodic set of earth departure opportunities and corresponding lunar space station approach opportunities to exist it is necessary that the opportunities occur at a repetitive set of inertial locations in the lunar orbital plane. This will happen if the ascending node of the earth departure orbit regresses one full rotation in a duration equal to some multiple of lunar months (one lunar month is 27.3 days), i.e.,

$$27.3\ n\ \omega_N = 360° \tag{1}$$

For example, for repetitive opportunities with the same station approach geometry on two month centers, equation (1) gives an earth orbit regression rate (ω_N) of 6.6°/day. From Figure 2, this regression rate can be provided by a 270 nm altitude, 30° inclination space station orbit; an inclination lying within the existing range of permissible launch azimuths at KSC (90°±18°). For a fixed altitude, azimuths in this range tend to maximize launch vehicle payload because an easterly launch takes maximum advantage of the earth's rotational peripheral velocity. Thus larger payloads can be transported from earth to the earth orbital facility than if the facility were located in an orbit of higher inclination.

Periodic opportunities separated by a larger number of months, say n = 3 or 4, would result in a decreased nodal regression rate requirement. The earth's departure orbit inclination and possibly altitude would need to be increased to provide the lower regression rate needed.

From Figure 3, each opportunity within the two-month period must satisfy the relationship:

$$\theta = \phi(t) + m\pi \ , \ m = 0,1. \tag{2}$$

where:

θ = angular location of earth-moon line with respect to the line of nodes of the lunar orbital plane with the earth's equatorial plane (deg).

ϕ = angular location of intersection of planes with respect to the line of nodes of the lunar orbital plane with the earth's equatorial plane (deg.).

Applying the law of cosines to the spherical triangle ABC in Figure 1 results in,

$$\alpha = \cos^{-1}\left[\cos i_L \cos i_o + \sin i_L \sin i_o \cos\omega_N t\right] \tag{3}$$

and by the law of sines

$$\phi(t) = \sin^{-1}\left[\frac{\sin i_o \sin\omega_N t}{\sin\alpha}\right] \tag{4}$$

where:

α = Angle between lunar orbital and earth departure planes (deg).

i_L = 22°, average inclination of the lunar orbital plane from 1975 to 1985, Reference 3.

i_o = 30°, inclination of earth departure orbit plane.

ω_N = -6.6°/day, earth departure orbit nodal regression rate for n = 2.

t = time (days).

Figure 4 presents $\phi(t)$ as determined by equations (3) and (4) for repetitive opportunities with a two-month period; i.e., $n = 2$. Also shown on the figure is $\theta = 13.2\ t + 60°$. The six points of intersection of θ and ϕ indicated in Figure 4 uniquely define the times of all the opportunities over the two-month period. The change in times of opportunities over the ten-year period from 1975 to 1985 due to the variation in inclination of the lunar orbital plane is negligibly small and therefore ignored.

The selection of an inertial angular location of 60° (Figure 3) for the initial opportunity is arbitrary and intended only to serve as an example for the purposes of this study. Any other angle could have been chosen, simply resulting in a translation of the linear relationship for θ and a corresponding shift in the times of the six opportunities. The number of opportunities in the two-month period can be neither increased nor decreased by changing the angular location of the initial opportunity.

The earth departure and arrival opportunities described above exist for only an instant of time, i.e., when the target earth-moon line passes through the intersection of the earth orbit and lunar orbit planes. Figure 5 illustrates how translunar and transearth launch windows of finite duration are obtained. The inertially fixed angular locations of two opportunities are indicated in the sketch. If upper and lower translunar flight time limits t_{TL2} and t_{TL1} are assumed then the translunar launch window duration is merely the difference of the two limits, i.e.,

$$LW_{TL} = t_{TL2} - t_{TL1} \tag{5}$$

Early in the window a slow translunar trajectory would be used allowing just enough time for the moon to move into the inertially fixed angular location of the opportunity for lunar arrival of the shuttle. Conversely, late in the window a fast translunar trajectory is used providing correspondingly less time for the moon to assume the correct angular location for the shuttle's arrival. However, as illustrated in Figure 6, the use of different flight time translunar trajectories (t_{TL2}, t_{TL1}) to achieve an earth departure window results in different lunar approach angles (γ_2, γ_1) relative to the earth-moon line, as defined by the velocity vector (V_∞) at the lunar sphere of influence, Reference 4. The slower the trajectory, the larger the lunar approach angle. Since the shuttle must rendezvous with the lunar space station located in an inertially fixed polar orbit, a shuttle plane change will be necessary except in the case when the approach angle coincides exactly with the station orbit ascending node. It is possible that some division of the total plane change requirement between maneuvers at both the earth and moon could result in a slightly lower total mission velocity requirement than a mission profile in which all of the plane change is executed at the moon. Investigation of this possible optimization is beyond the scope of this study. The strategy for rendezvous with the station for a minimum total velocity change expenditure involving a plane change maneuver only at the moon is described in Section 4.2.

Similarly, the lunar departure window is obtained through the use of different total transearth flight times, i.e.,

$$LW_{TE} = t_{TE2} - t_{TE1} \tag{6}$$

A shuttle plane change is also required for lunar departure in order to provide the correct lunar departure angle relative to the earth-moon line for the velocity vector when it emerges from the lunar sphere of influence. The transearth maneuver is described in Section 4.3.

4.0 MISSION VELOCITY REQUIREMENTS

The purpose of this section is to determine the characteristic total velocity requirement of a round trip lunar logistic mission for the shuttle. The TLI velocity change is determined using an iterative technique to include gravity losses. The LOI and TEI velocity change requirements are determined as a function of flight time; shuttle LOI and TEI velocity capabilities are arbitrarily selected thereby determining the duration of the translunar and transearth launch windows for each opportunity. The EOI velocity change is assumed to be the same as that for a TLI maneuver at the same altitude except that gravity losses are neglected because of the relatively short duration (high thrust/weight) of the maneuver.

4.1 Translunar Injection (TLI)

The velocity change not including gravity losses for the shuttle translunar injection maneuver is about 10,192 fps. This value changes slightly as a function of translunar flight time but the variation is small enough to be considered negligible. Gravity loss is presented in Figure 7 as a function of shuttle gross weight (not including translunar payload) and is conservative since it represents shuttle departure from a 100 nm rather than a 270 nm orbit. The data is based on a shuttle with a single 75 K lb thrust NERVA engine with an I_{sp} of 800 sec. (Reference 5).

4.2 Lunar Orbit Insertion (LOI)

In general, the shuttle enters the lunar sphere of influence with an approach angle which makes a plane change maneuver necessary for rendezvous with the lunar orbital station. This is accomplished for minimum velocity change by using a three impulse LOI maneuver which is illustrated in Figure 8. The first of the three maneuvers is a non-periapse velocity change which transfers the shuttle from its lunar approach hyperbola to an elliptic orbit whose line of apsides is colinear with the lunar polar axis, and whose periselene altitude is slightly higher than the station orbital altitude. The necessary plane change is executed at the first aposelene and at periselene the shuttle transfers to a slightly higher altitude circular orbit which is coplanar with the station orbit. When the correct phasing with the station is attained the shuttle performs a Hohman transfer immediately followed by terminal rendezvous. The total velocity requirements for the three impulse LOI maneuver as a function of longitude of the station orbit ascending node and translunar flight time have been determined in Reference 6. The Hohman transfer and terminal rendezvous maneuvers are negligible in comparison to the three impulse LOI velocity change.

Figures 9A and 9B present the three impulse velocity requirements as a function of translunar flight time for the six earth departure and arrival opportunities. These data are based on an inertially fixed station polar orbit whose ascending node is located 60° west of the earth-moon line at the first opportunity of the two month period.* Selection of a reasonable upper time limit for translunar flight time of 84 hours (3.5 days) establishes one boundary of the earth departure windows. As indicated in Figures 9A and 9B an LOI velocity capability of 3600 FPS provides five earth departure launch windows during the two month period varying in duration from 3 hours to 24 hours. The duration of the launch window for each earth departure opportunity is given in Table 1.

4.3 Transearth Injection (TEI)

The TEI maneuver is also a three impulse procedure aimed at providing the correct lunar departure angle for the shuttle velocity vector at lunar escape. It is the same as the LOI maneuver with the phasing Hohman transfer eliminated and the order of occurrence of the three impulses reversed.

Figures 10A and 10B present the TEI velocity change requirements as a function of total shuttle flight time for the

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*It is assumed that the lunar gravitational field does not cause precession of the polar orbit.

six opportunities. Total flight time is measured from the instant the shuttle departs from the lunar orbit station and consists of time spent in the plane change ellipse as well as the transearth flight time. The lunar departure window duration must be determined from total flight time because the period of the plane change ellipse varies depending on the transearth flight time. Thus the variable time spent in the plane change ellipse as well as the variable transearth flight time determine the window duration. This is not the case in determining earth departure windows because there is no plane change ellipse prior to the TLI maneuver.

The upper bound on the lunar departure window is also determined by an 84 hour (3.5 days) flight time constraint on the transearth trajectory corresponding to 120 hours of total flight time because of the 36 hour plane change ellipse. As indicated in Figures 10A and 10B, the chosen TEI velocity capability of 3300 fps provides four lunar departure windows during the two month period with durations varying from 6 hours to 35.5 hours. The duration of the lunar departure window corresponding to each opportunity is given in Table 1.

4.4 Earth Orbit Injection (EOI)

At earth arrival the shuttle maneuvers into a 300 nm circular phasing orbit. The retrograde velocity change requirement is assumed to be the same as that for a translunar injection from a 300 nm orbit, i.e., 10,150 fps. Gravity losses are neglected because of the relatively high shuttle thrust to weight ratio and therefore short duration of the EOI maneuver. When phasing with the earth orbital facility in the 270 nm orbit is correct, the shuttle performs a Hohman transfer to the lower orbit, and immediately rendezvous' with the facility. The velocity requirement for the transfer is 253 fps resulting in a total EOI velocity requirement of 10,403 fps.

4.5 Total Velocity Requirement

The translunar velocity requirement of 14,452 fps is obtained by adding the individual TLI and LOI velocity changes.* Addition of the TEI and EOI velocities results in a transearth velocity requirement of 13,703 fps. These velocity expenditures provide 5 earth departure windows varying in duration from 3 hours to 24 hours and 4 lunar departure windows from 6 hours to 35.5 hours in duration. Each window is available every two months.

5.0 VEHICLE SIZING

The cislunar shuttle is assumed to be a nuclear propelled vehicle with an average Isp of 795 seconds (which includes

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*The TLI gravity loss of 660 fps was determined by iteration using the data presented in Figure 7.

reactor cooldown after each maneuver) and a mass fraction of .75 (Reference 5). According to Reference 7, the baseline translunar payload is 119 K lbs with zero payload returned to earth.

Using the mission velocity changes derived in Section 4 and the rocket equation in an iterative mode to determine the TLI gravity loss (660 fps) the gross weight of the shuttle is determined to be 352 K lbs. The total vehicle weight at the beginning of the mission is 471 K lb of which 119 K lbs is translunar payload; 264 K lb is liquid hydrogen propellant and 88 K lb is shuttle dry weight.

7.0 CONCLUSIONS

To illustrate the lunar logistics portion of an integrated manned space flight program, a representative lunar logistics mission mode providing repetitive earth departure and lunar departure windows with a two month period has been derived. It was found that this particular mode is compatible with the earth orbital station being located in a 270 nm altitude orbit with an inclination of 30°.

The translunar shuttle is sized to give sufficient velocity change capability to provide 5 earth departure windows varying in duration from 3 hours to 24 hours and 4 lunar departure windows of 6 hours to 35.5 hours in duration. Each of the windows occurs every two months.

The vehicle is nuclear propelled with an average Isp of 795 seconds (considering reactor cooldown after each maneuver) and a mass fraction of .75 is assumed. Based on the required mission velocity changes and baseline translunar and transearth payloads of 119 K lbs and 0, respectively, the translunar shuttle gross weight is 352 K lbs.

E. M. Grenning

1013-EMG-ajj   E. M. Grenning

Attachments

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3. J. O. Cappellari, Jr., "A Compendium of the Moon's Motions and Geometry: 1966 Through 1985," Bellcomm TR-68-310-1, March 29, 1968.

4. J. A. Herbaugh, et al, "Lunar Orbit Rendezvous Trajectory Data Package (U) - Apollo Mission Parameter Analysis," T.R.W., 8408-6050-RC-000, May 29, 1964.

5. Private conversations with D. J. Osias and A. L. Schreiber, Bellcomm, Inc.

6. LMSC, "Lunar Orbital Survey Missions - Final Report, Vol. I Summary," Contract NAS9-5288, January 16, 1967.

7. LMSC, "Nuclear Flight System Definition Study, Phase II," Contract NAS8-24715, May 1, 1970.

TABLE 1. TIME BETWEEN OPPORTUNITIES AND LAUNCH WINDOW DURATIONS FOR TWO MONTH PERIODIC LUNAR LOGISTICS SCHEDULE.

OPPORTUNITY	TIME BETWEEN OPPORTUNITIES (DAYS)	TLI LAUNCH WINDOW (HRS) (ΔV_{LOI} = 3600 FPS)	TEI LAUNCH WINDOWS (HRS) (ΔV_{TEI} = 3300 FPS)
1	10.2	24	0
2	10.35	0	0
3	10.00	20	15
4	8.95	20	35.5
5	5.4	19	32.5
6	9.7	3	6
1			

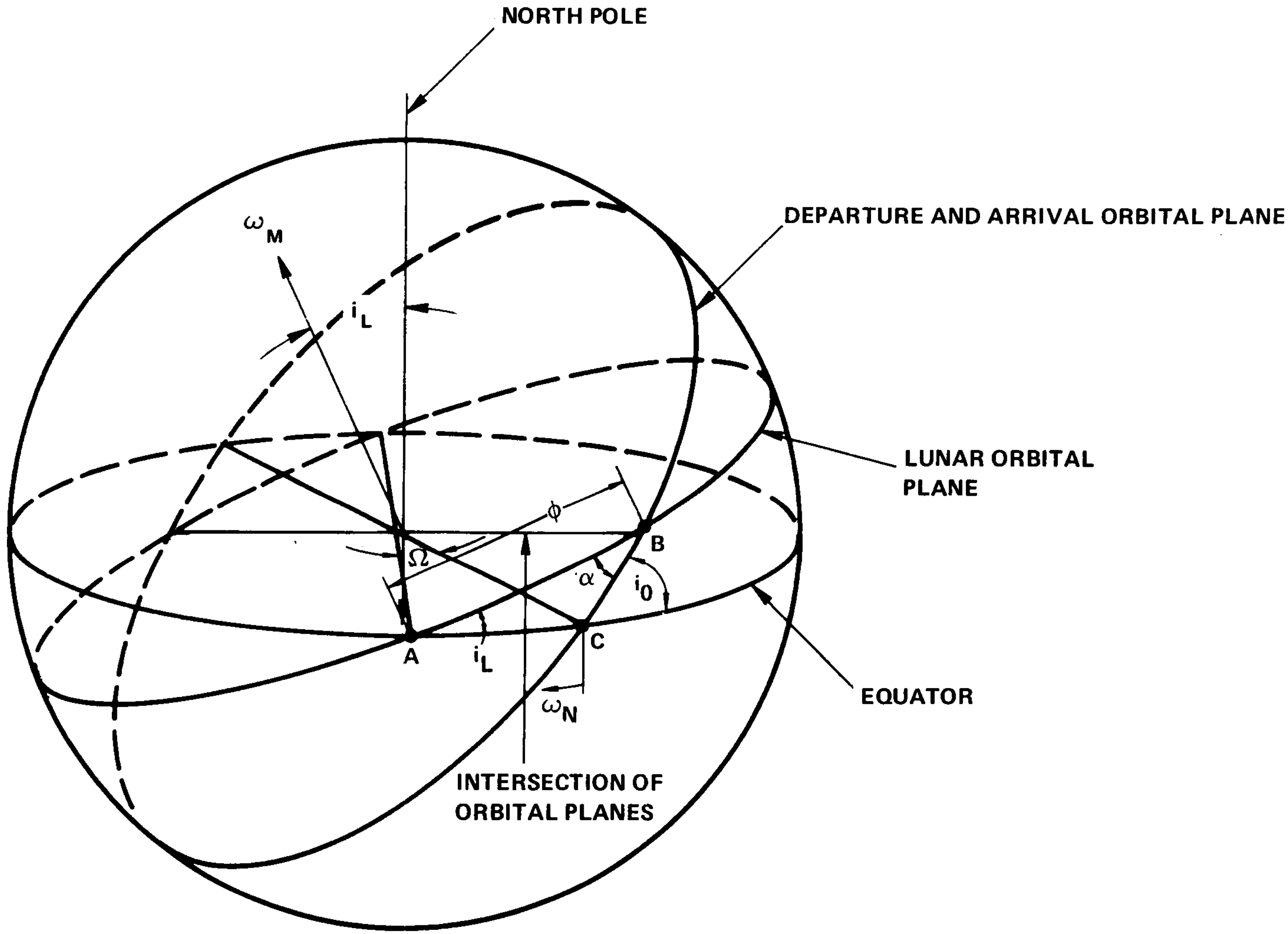

FIGURE 1 - EARTH DEPARTURE AND ARRIVAL GEOMETRY

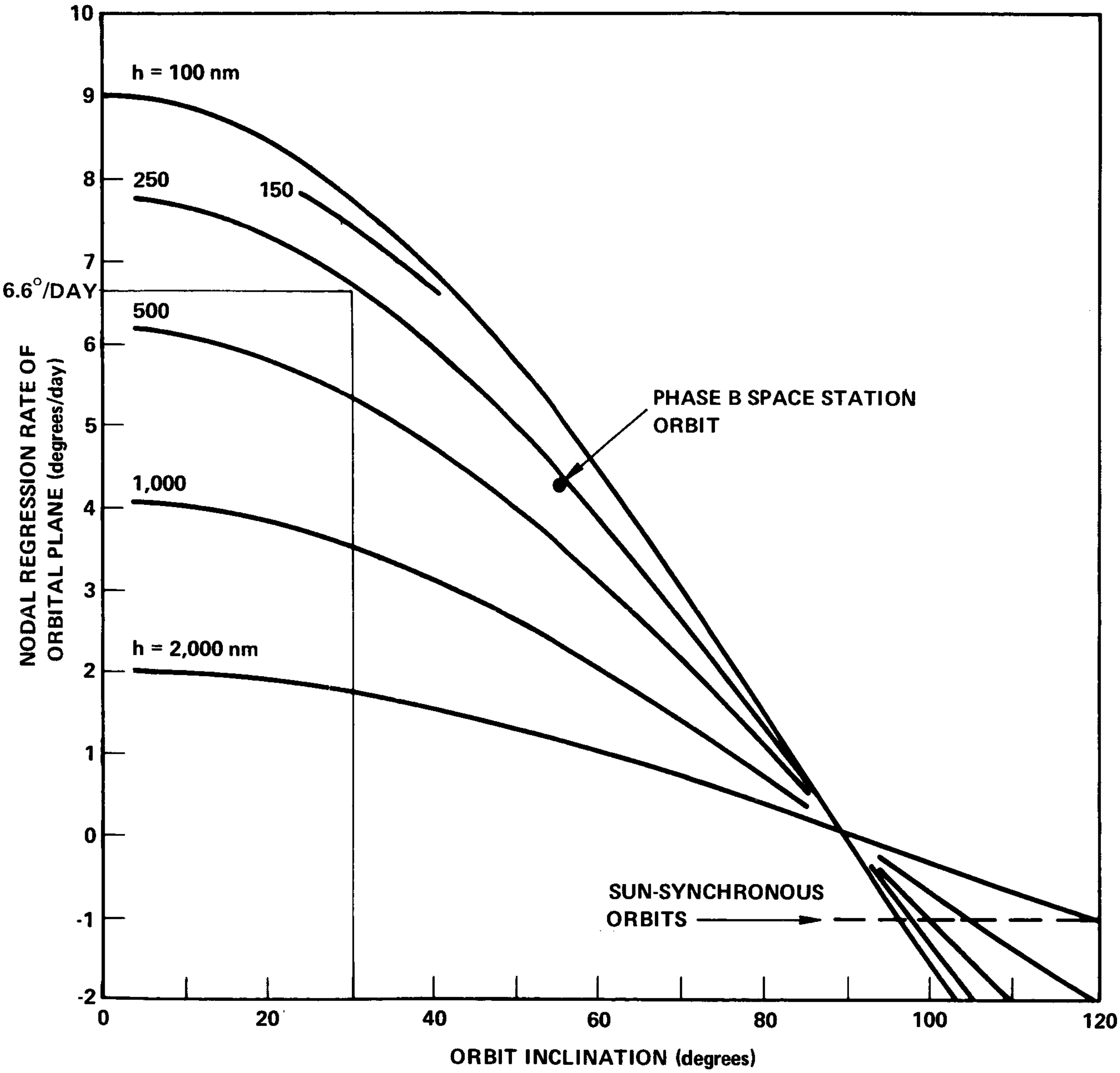

FIGURE 2 - ALTITUDE AND INCLINATION EFFECTS ON NODAL REGRESSION RATE OF CIRCULAR ORBITS

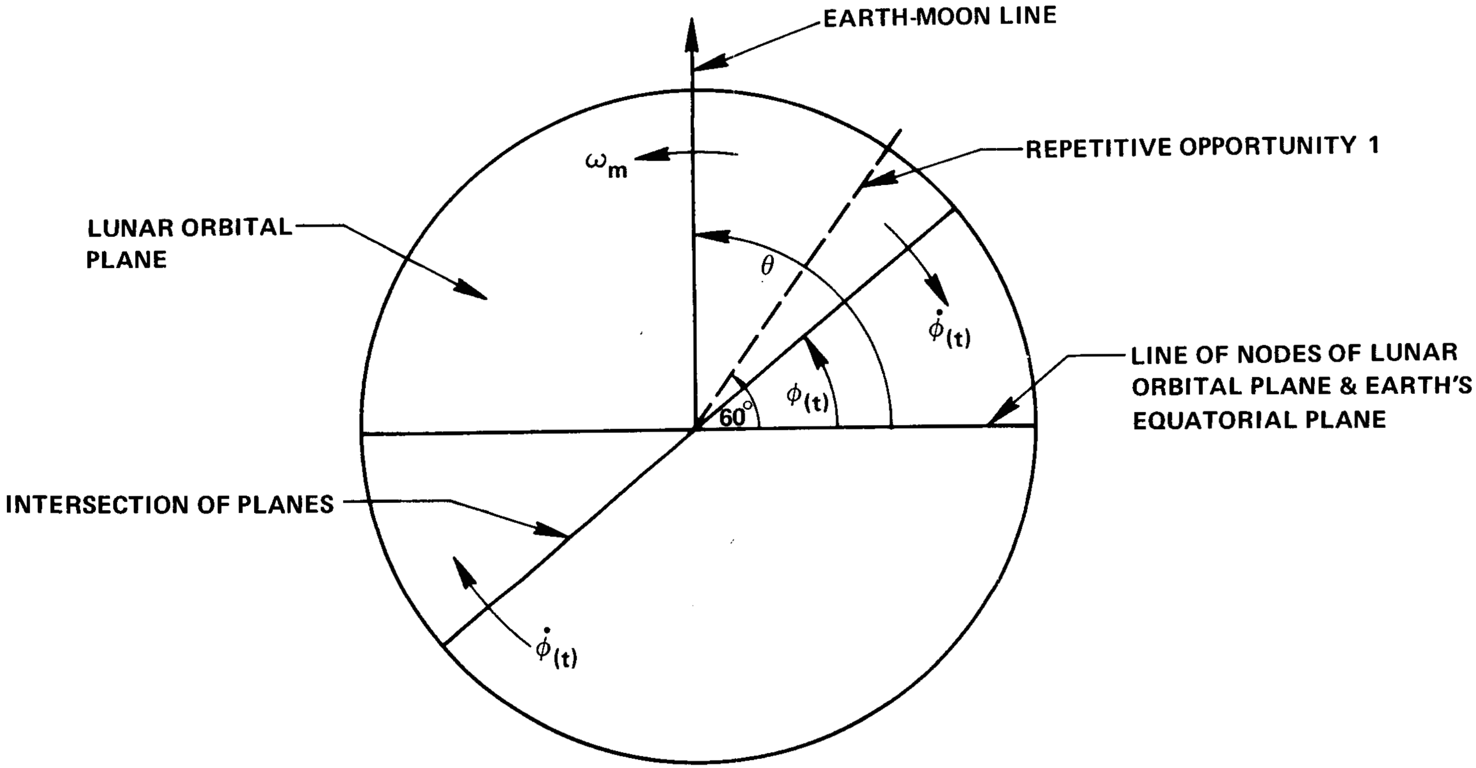

FIGURE 3 - MOTION OF EARTH-MOON LINE AND LINE OF INTERSECTION IN LUNAR ORBITAL PLANE

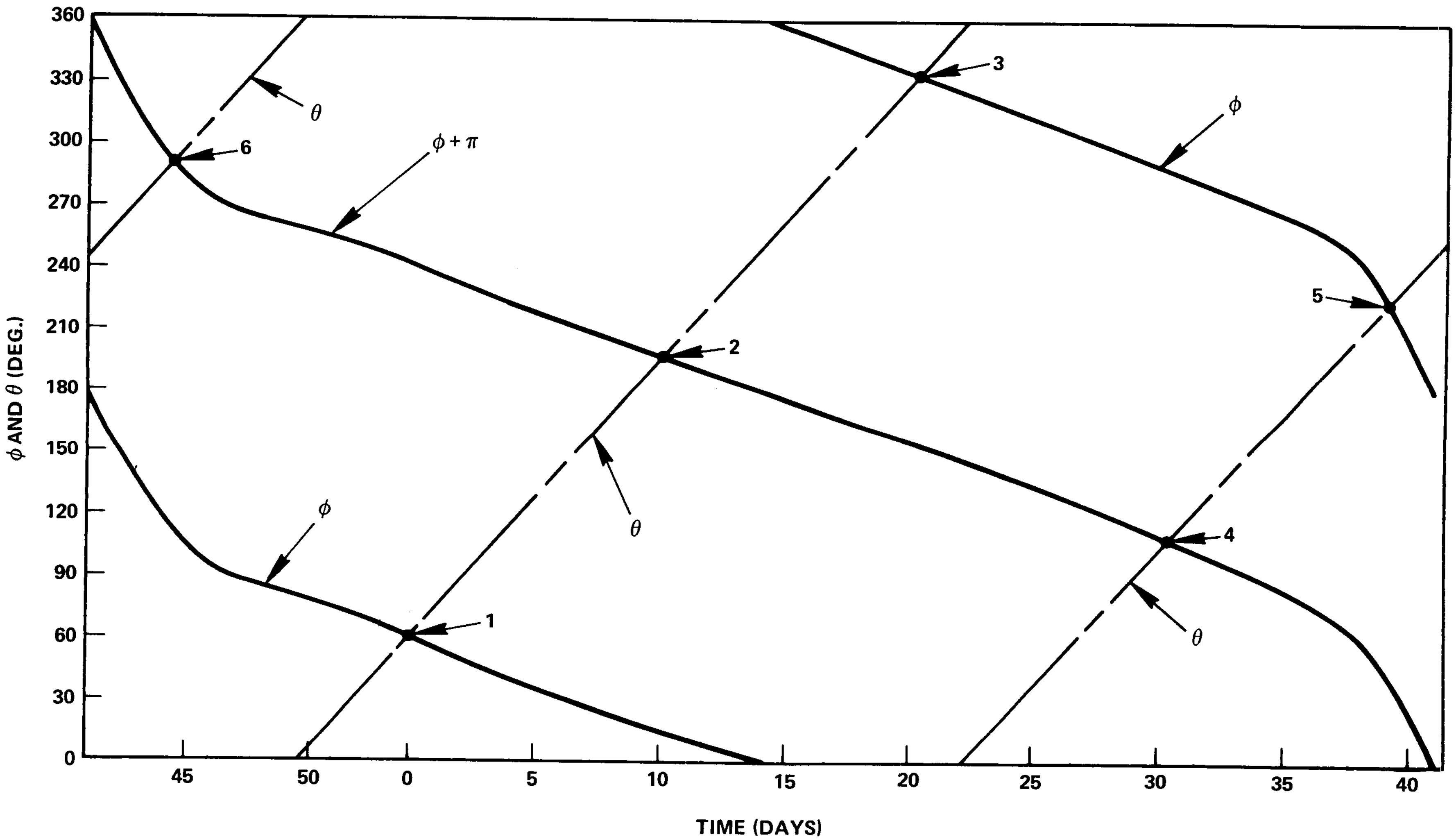

FIGURE 4 - ANGULAR LOCATION OF INTERSECTION OF PLANES (ϕ) AND EARTH-MOON LINE (θ) AS A FUNCTION OF TIME FOR n = 2 MONTHS

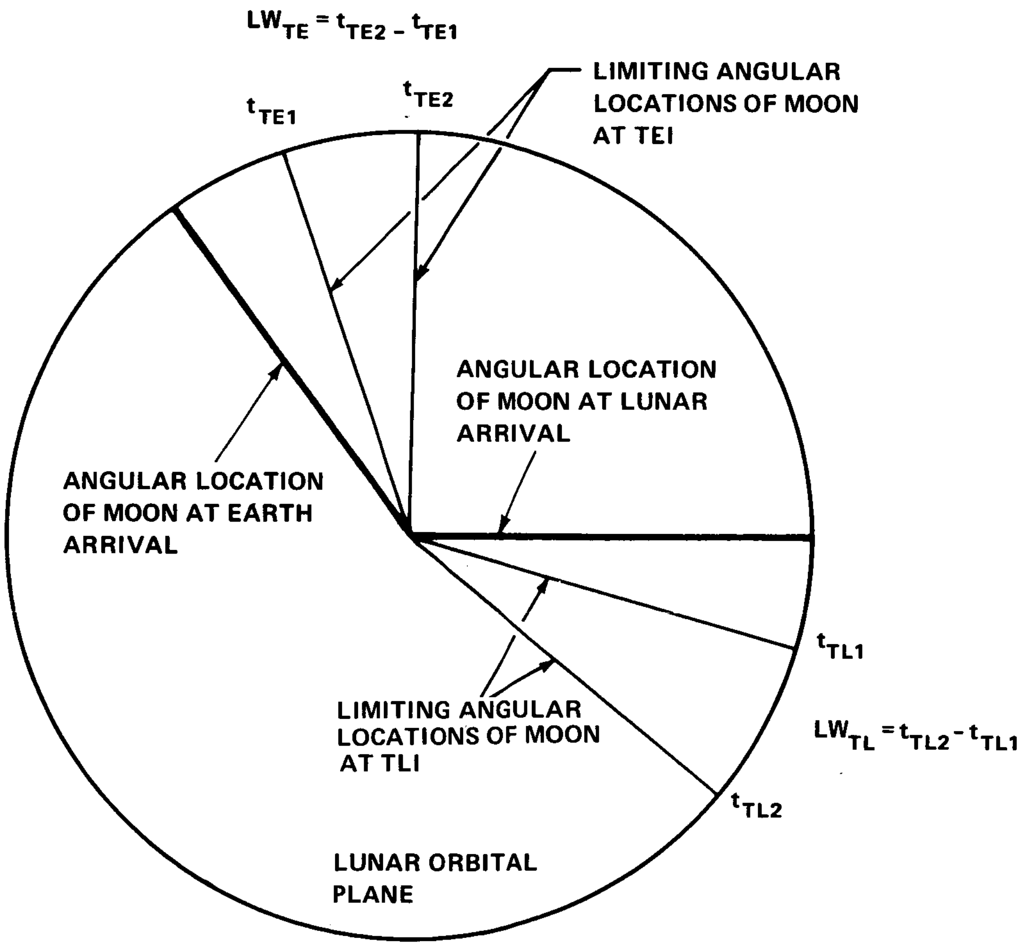

FIGURE 5 - TRANSLUNAR AND TRANSEARTH LAUNCH WINDOW DURATIONS

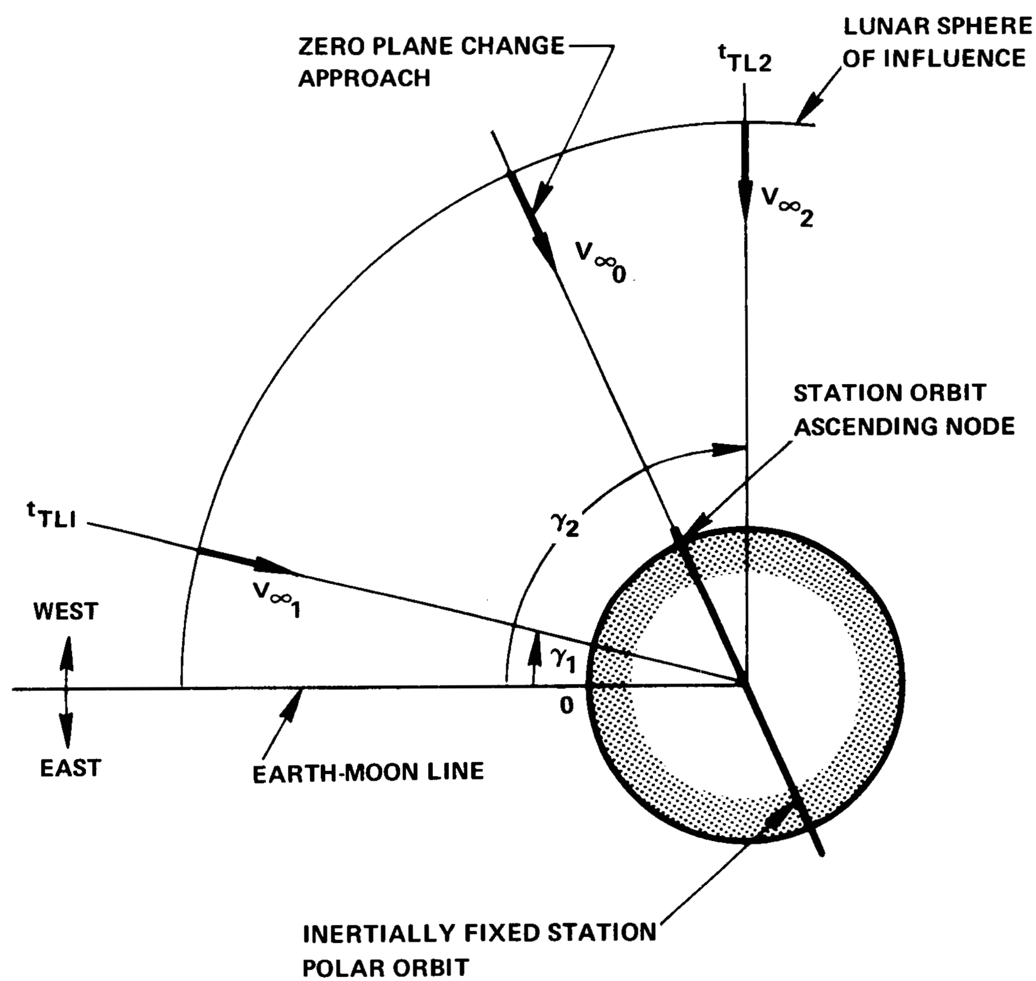

FIGURE 6 - LUNAR APPROACH GEOMETRY

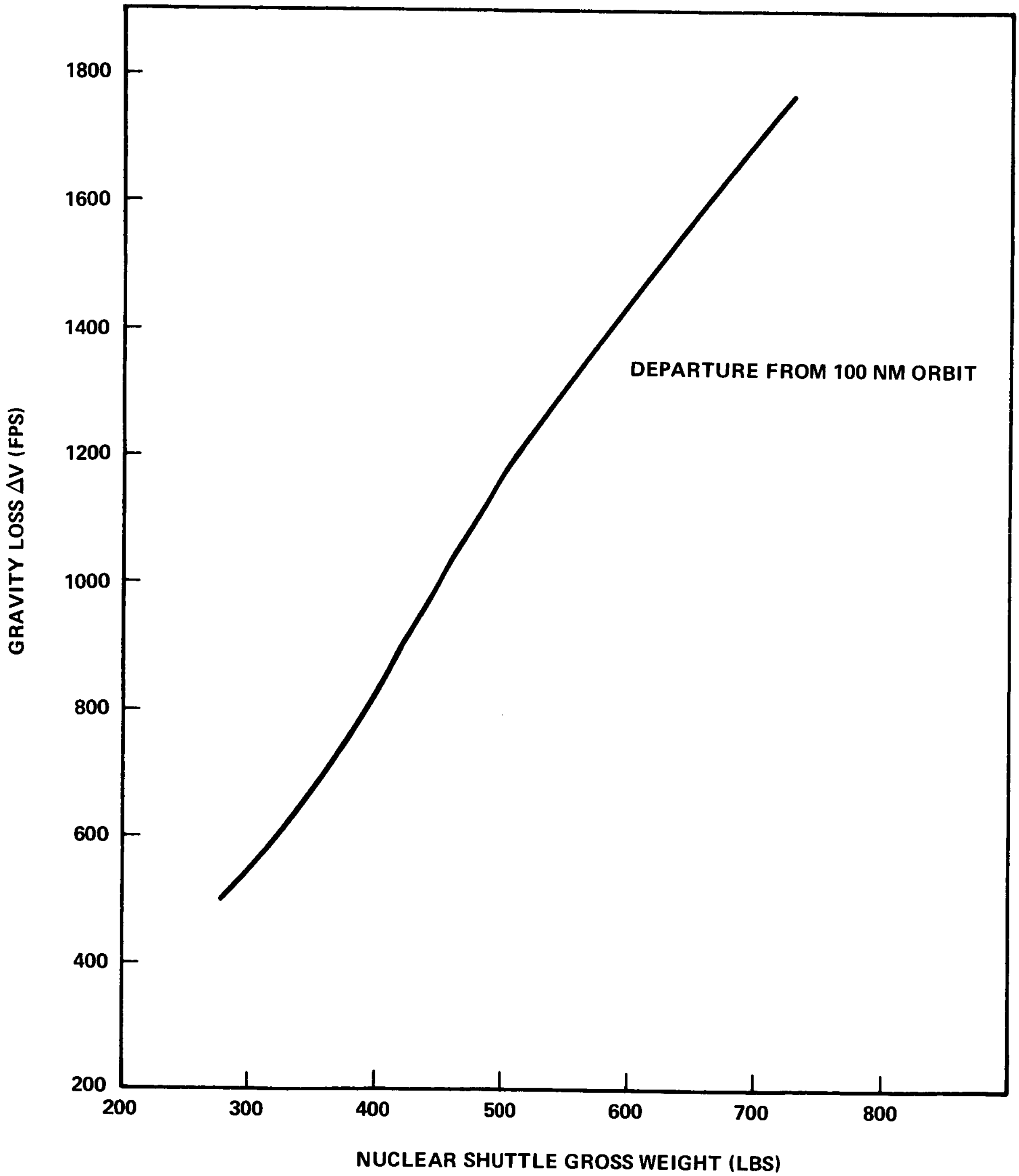

FIGURE 7 - GRAVITY LOSS ΔV VS NUCLEAR SHUTTLE GROSS WEIGHT

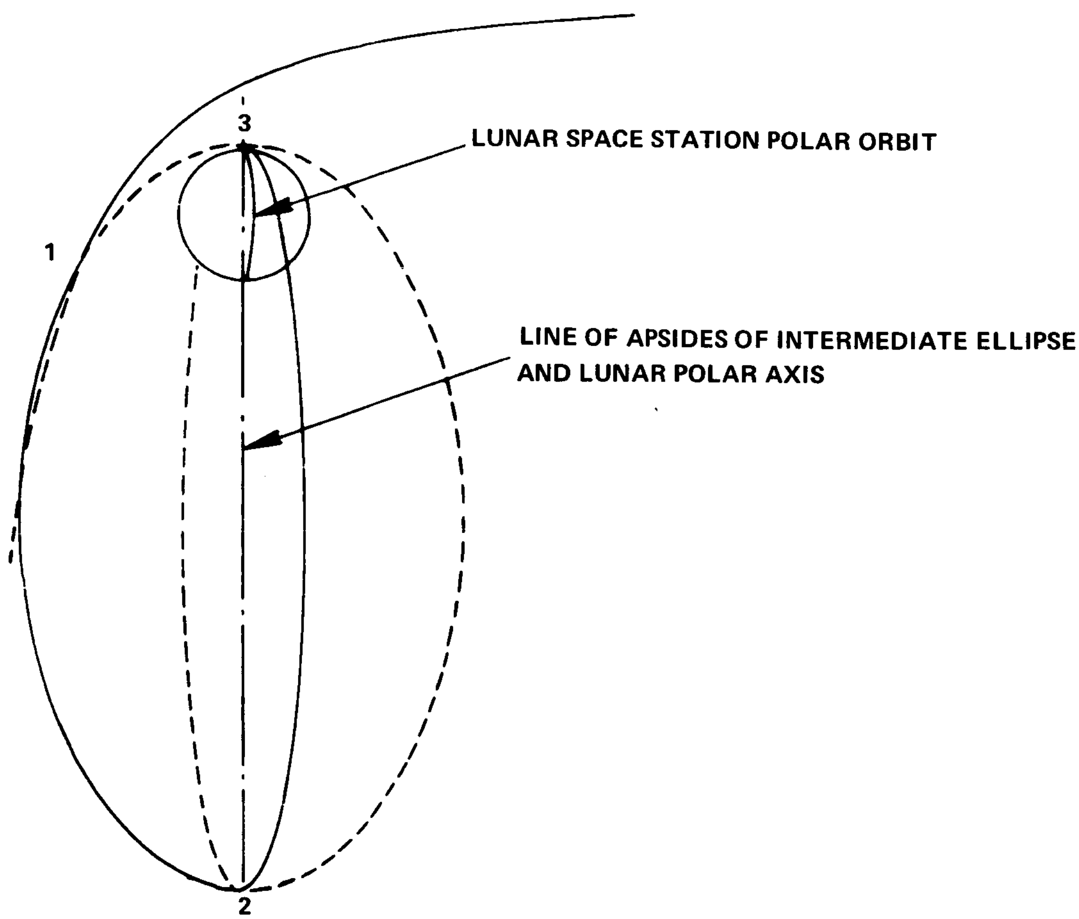

FIGURE 8 - THREE-IMPULSE GEOMETRY

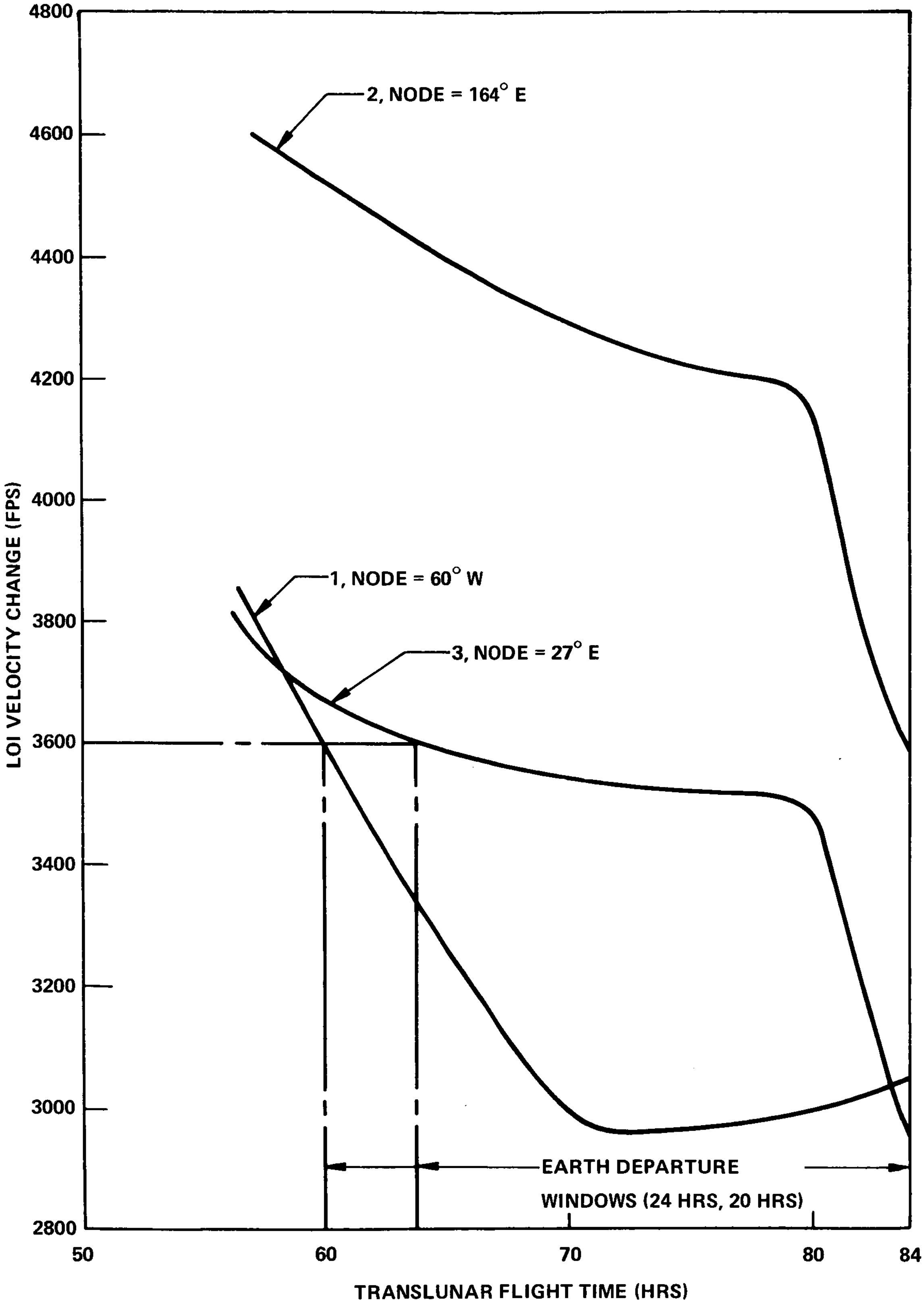

FIGURE 9A - THREE IMPULSE LOI VELOCITY CHANGE VS TRANSLUNAR FLIGHT TIME – OPPORTUNITIES 1, 2, & 3

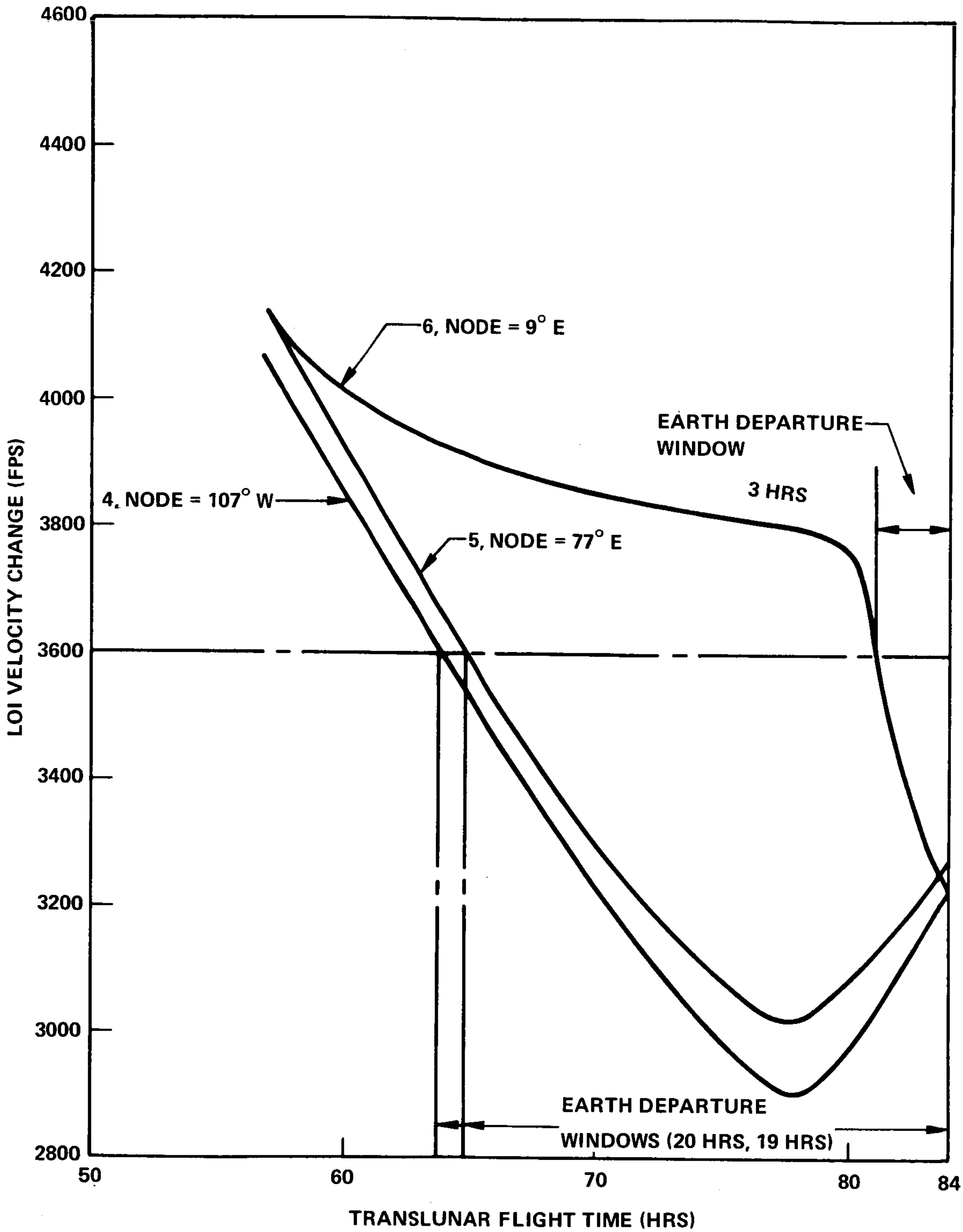

FIGURE 9B - THREE IMPULSE LOI VELOCITY CHANGE VS TRANSLUNAR FLIGHT TIME – OPPORTUNITIES 4, 5, & 6

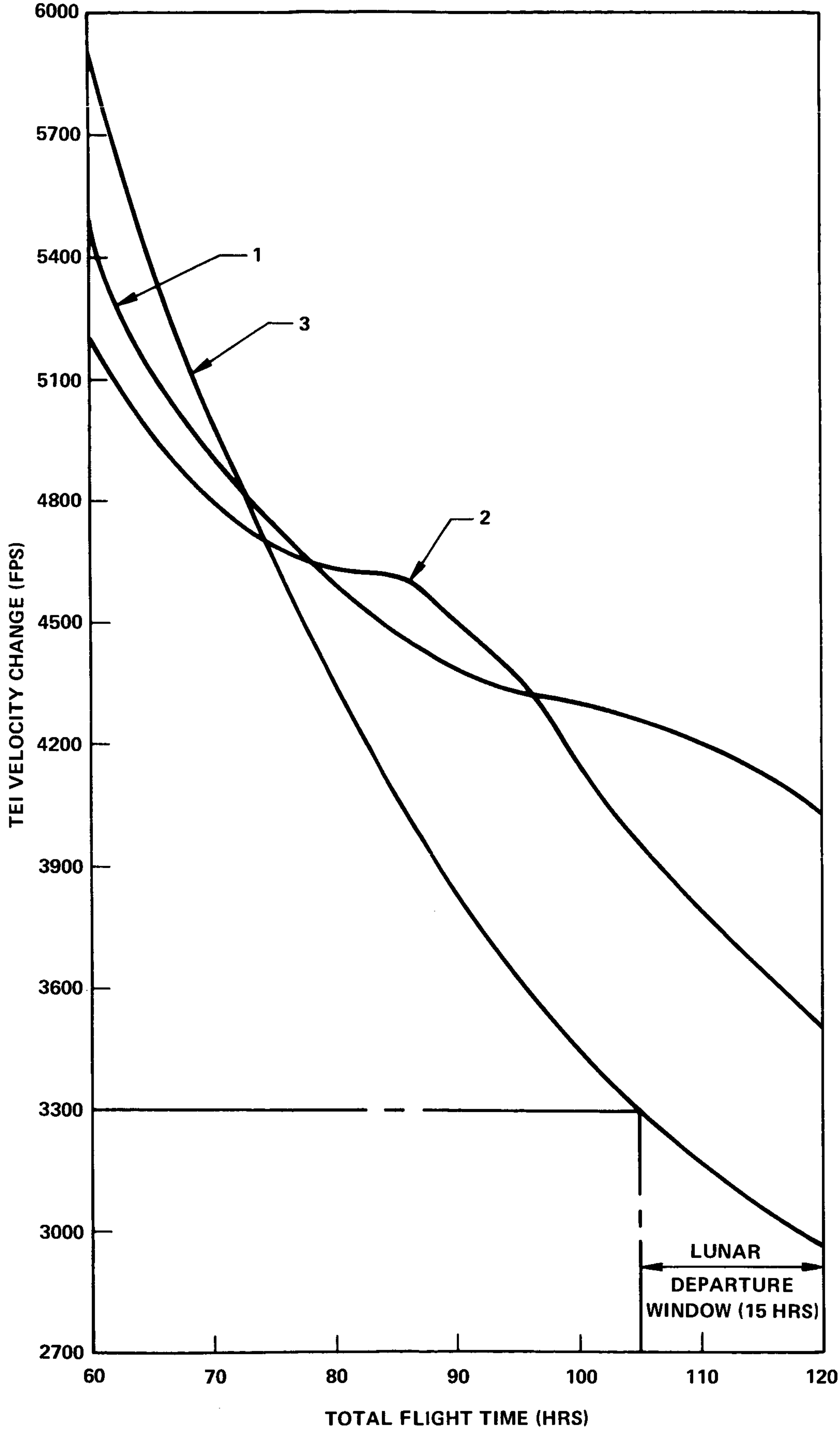

FIGURE 10A - THREE IMPULSE TEI VELOCITY CHANGE VS TOTAL FLIGHT TIME – OPPORTUNITIES 1, 2, & 3

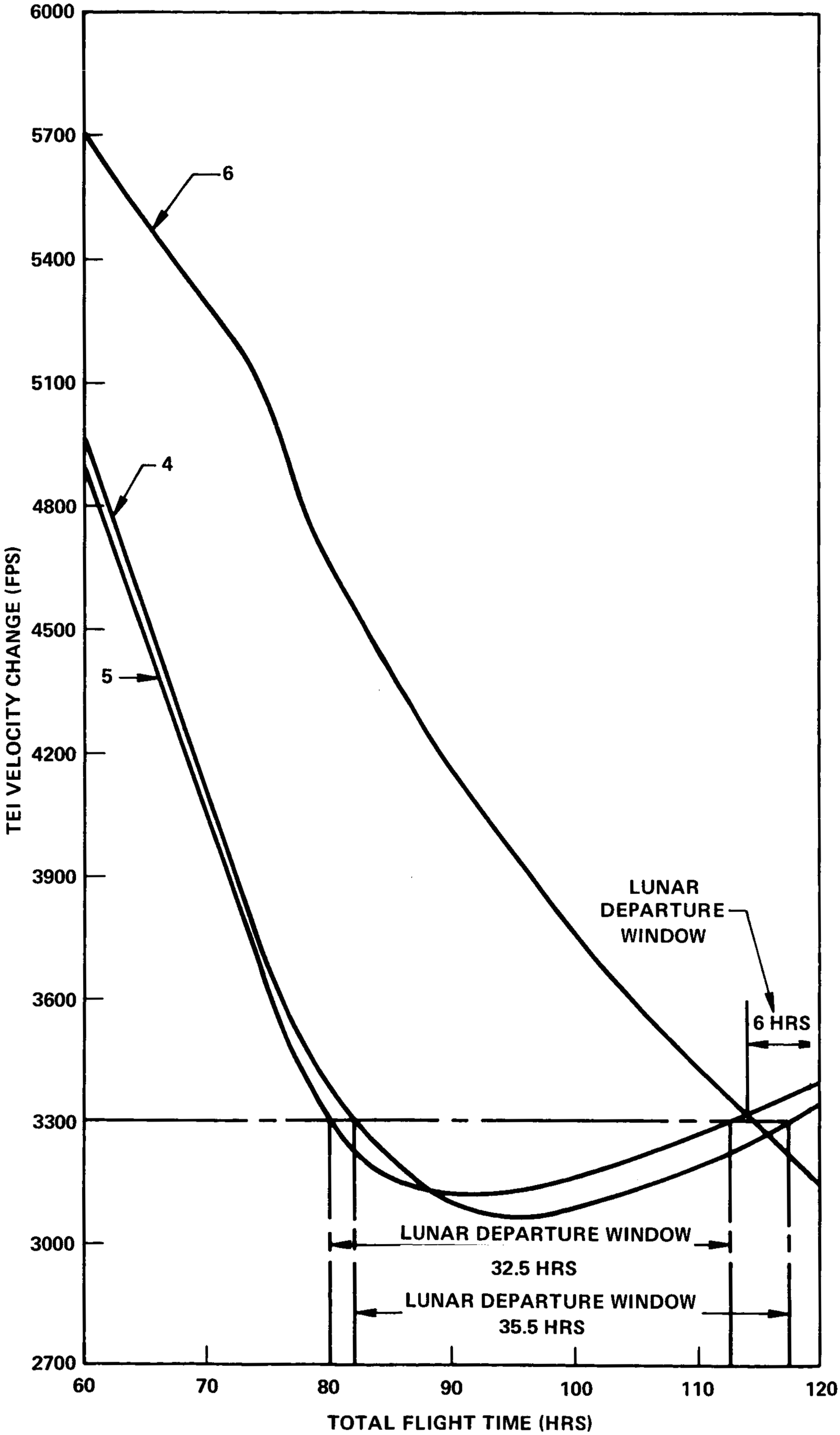

FIGURE 10B - THREE IMPULSE TEI VELOCITY CHANGE VS TOTAL FLIGHT TIME — OPPORTUNITIES 4, 5, & 6